NACA TN 3486

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3486

MEASUREMENTS OF TURBULENT SKIN FRICTION ON A FLAT PLATE AT TRANSONIC SPEEDS

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Washington

September 1955

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SUMMARY

The present report describes the design and construction of a floating-element skin-friction balance. This instrument, which is essentially an improved version of Dhawan's balance, was applied to measurements of local skin friction in the turbulent boundary layer of a smooth flat plate at high-subsonic Mach numbers and supersonic Mach numbers up to 1.75. The measured skin-friction coefficients are consistent with the results of other investigations at subsonic and also at supersonic speeds. The principal difficulties which exist in comparing skin-friction coefficients at various Mach numbers are discussed.

INTRODUCTION

In recent years the importance of viscous effects in modern aerodynamics has motivated a great number of both theoretical and experimental investigations in the field of boundary-layer phenomena. For obvious reasons, particular attention has been given to the frictional forces introduced by these effects on the surfaces of bodies moving in a fluid. For the laminar boundary layer the theoretical evaluation of these forces has reached a well-advanced stage, whereas for the turbulent boundary layer the lack of knowledge of even some of the most basic properties of turbulent shear flows has prevented the development of quantitative theories. Thus, the skin friction associated with the turbulent boundary layer presents an important problem for experimental research.

There are many different ways of measuring skin friction (an exhaustive discussion of these is given in ref. 1). However, it is clear that a direct measurement of the frictional force itself would be preferable to all others, provided that it could be performed with satisfactory accuracy. This technique was first used by Kempf in 1929 (ref. 2) and then by Schultz-Grunow in about 1940 (ref. 3). It was used again in 1950 by Dhawan (ref. 1), who developed and demonstrated a satisfactory, relatively simple, small, and compact skin-friction meter adaptable to high-speed flow. Dhawan's work gave rise to a number of parallel investigations at different values of the flow parameters with instruments based on the principles of his original device. Some of the results have already been published. Coles (ref. 4) and Bradfield, DeCoursin, and Blumer (ref. 5) have dealt with flat plates at Mach numbers from 2 to 5 and with axially symmetric configurations, respectively.

While Dhawan was able to obtain reliable measurements of subsonic skin friction on a flat plate with his original instrument, he encountered difficulties in extending his experiments to the supersonic range in the GALCIT 4- by 10-inch transonic wind tunnel. The present investigation was undertaken in order to remedy this situation and to develop an instrument capable of reliable measurements of skin friction at supersonic speeds, even under the restrictions imposed by the size and characteristics of the existing facilities.

In the course of the investigation it was realized that problems of a more fundamental nature than the limitations of geometrical size or the establishing of satisfactory flow conditions were involved in the measurements and in their interpretation. The lack of knowledge about the proper definition of an "origin," that is, of an absolute streamwise Reynolds number scale of a turbulent boundary layer, makes it difficult to find a satisfactory way of comparing skin-friction coefficients of low- and high-speed turbulent boundary layers in a unique manner. This problem, which is of major importance at the Reynolds numbers obtained in the present experiments (approximately 10^6, based on the distance between leading edge and point of measurement), is discussed in detail in this report. A great deal of effort has been expended in studying the characteristics of transition observed in the experiments as well as in the effects of various tripping devices.

In addition, one of the purposes of the present investigation was to develop an absolute instrument for use as a calibration device for other means of measuring skin friction currently under development at the Guggenheim Aeronautical Laboratory of the California Institute of Technology.

This investigation was conducted at GALCIT under the sponsorship and with the financial assistance of the National Advisory Committee for Aeronautics.

SYMBOLS

c_f local skin-friction coefficient, τ_w/q

m ratio of trip mass flow per unit width of flat plate to mass-flow defect of laminar boundary layer, $\rho_\infty U\delta^*$

M	Mach number
p	static pressure
p_τ	pressure indicated by total head probe resting on probe surface
q	free-stream dynamic pressure, $\rho_\infty U^2/2$
R	Reynolds number
R_x	Reynolds number based on x, $U\rho_\infty x/\mu_\infty$
R_y	Reynolds number based on y, $U\rho_\infty y/\mu_\infty$
R_θ	Reynolds number based on θ, $U\rho_\infty \theta/\mu_\infty$
u	x-component of local velocity
U	free-stream velocity outside boundary layer
x	streamwise coordinate
y	coordinate normal to surface
γ	ratio of specific heats, 1.4 for present investigation
δ^*	boundary-layer displacement thickness
θ	boundary-layer momentum thickness
μ	viscosity
ρ	density
τ	shear stress

Subscripts:

i	incompressible flow
w	value at solid surface
∞	free-stream value outside boundary layer

INSTRUMENTATION AND GENERAL TECHNIQUE FOR MEASUREMENTS OF LOCAL SKIN FRICTION

The skin-friction meter used in the current investigation is a direct development of the original Dhawan instrument (ref. 1) and differs only in structural details. (Figs. 1, 2, and 3 show the instrument with modifications for the supersonic experiments; in the subsonic experiments the leading edge was 1.2 centimeters closer to the measuring element and no trip air holes existed.) The skin-friction force acts on an elastically supported floating element in the plane of the flat plate and causes a streamwise deflection of this element. The displacement creates an electric signal which can easily be interpreted in terms of the skin-friction force.

The flat plate was placed horizontally in the test section, approximately halfway between the ceiling and the floor, and fastened to the test-section side windows at four points (fig. 3). All structural members of the mechanism were attached directly to the plate and housed within a sealed windshield. The instrument forms a compact unit which can be completely assembled and calibrated outside the tunnel, thus permitting a very simple and quick installation. The angle of incidence of the flow over the instrument can be adjusted by rotating the windows, and the streamwise position of the instrument can be changed by interchanging the removable side wall panels of the test section. By means of these adjustments the conditions of nonuniform flow, which had seriously interfered with the preliminary measurements of reference 1 in the same tunnel, were eliminated and pressure distributions within $\Delta p/q = \pm 0.02$ were realized throughout the whole range of experiments.

Techniques used in fabrication and adjustment of the instrument were essentially those described in reference 1. Alinement of the floating element with the fixed surface was checked optically before and after each series of experiments and, in addition, the repeatability of a few subsonic measurements was ascertained each time. Although it is difficult to give an accurate numerical criterion, it is felt that under the conditions of the present measurements any misalinement must be kept within 10^{-4} inch in order to avoid noticeable systematic errors.

Force-Sensing System

A Schaevitz 040-L variable differential transformer was used as the displacement-sensitive element. The electrical circuit (fig. 4) is similar to the one discribed in reference 1. A Hewlett-Packard 200B oscillator was used as the power supply. The output circuit was equalized by a 25,000-ohm potentiometer. Input and output voltages were read on Hewlett-Packard vacuum-tube voltmeters. Input frequency was 20 kilocycles in all cases and the input voltage ranged from 4 to 10 volts root mean square. The sensitivity of the instrument is determined by the flexure links. At subsonic speeds flexure links made of 0.0045- by 1/16-inch steel strip gave, at 4-volt input, a sensitivity of 6.05 volts per centimeter or 0.144 volt per gram. At supersonic speeds the flexure linkage was made more rigid (0.006- by 1/16-inch) in order to withstand shock loads. A sensitivity of 15.1 volt per centimeter or 0.129 volt per gram resulted at 10-volt input. In both cases a small dashpot filled with silicone fluid of 1,000 centistokes viscosity provided adequate damping. Calibration was carried out by hanging weights directly on the moving part of the mechanism with a single-fiber nylon string passing over an aluminum pulley on jewel bearings. Calibration curves used for the present results are given in figure 5. The repeatability of calibration was within ±1 percent of the readings recorded in the high-subsonic and supersonic regions. This performance together with observations of zero shift discussed in the next section led to the conclusion that the added complexities of a null-reading system or of a more elaborate electrical circuit would not be justified. The possible, and by no means guaranteed, improvements in accuracy would be small because of the difficulties imposed by the small size of the instrument.

Effects of Thermal and Elastic Deformations

The most serious problem in the design of the instrument was the elimination of zero shift, which was due mainly to thermal deformation of the structural parts of the instrument and, to some extent, also of the wind tunnel. After a considerable amount of experimentation a satisfactory solution was obtained. It is believed that a homogeneous structure completely symmetrical about the point of measurement is essential to the success of this type of instrument. The importance of thermal deformations is apparent from the fact that the total travel of the measuring element, corresponding to the maximum measurable force, was 0.012 centimeter in the subsonic experiments and 0.005 centimeter in the supersonic ones. There was no noticeable thermal effect on the electrical equipment, as was shown by calibrations over the range of surface temperatures occurring in the tunnel. A Western Electric Thermistor was mounted in the plate near the force element in order to ascertain that thermal equilibrium had been reached before measurements were taken.

In order to eliminate all possible effects arising from the operation of the tunnel (thermal, elastic, vibrational), experiments were run at each flow condition with the element isolated from the flow by means of a sealed shield which could easily be attached to the plate. The results of these experiments showed that no zero-shift correction was necessary at subsonic speeds, where the surface of the instrument was approximately at room temperature. In the supersonic experiments the stagnation temperature had to be raised to 55° to 65° C in order to avoid condensation of water vapor in the test section. A zero shift of approximately 1 percent of the force at these speeds was then observed during the operation of the tunnel. It was, however, found to be repeatable if the tunnel was warmed up before a measurement was made and if the duration of the run was not longer than 5 or 10 minutes. With the same precautions, the initial and final zero readings taken immediately before and after an experiment were found to be in agreement within the accuracy of the calibration of the instrument.

In the experiments of Dhawan, the measured skin friction was strongly influenced by rate of change of stagnation temperature, this being tentatively attributed to the effect of heat transfer on the skin friction. The magnitude of this effect was up to 50 percent of the surface shear stress in a direction opposite to theoretical predictions. In the development of the present instrument, a preliminary model showed the same behavior, which, it was definitely established, was caused by transient thermal deformations of the instrument structure. The final instrument gave, under the same conditions, an effect of only a few percent in the direction predicted by theory. It cannot be concluded, however, that the remaining small effect is entirely due to heat transfer. Reliable information on this phenomenon should be obtained by instruments designed specifically for this purpose.

Pressure Distributions

Pressure distributions were measured by means of six pressure orifices on the plate. In addition, the gap around the element and the outlets for trip air at supersonic speeds were used as pressure orifices. In the adjustment of the pressure field, great care was taken to obtain not only a satisfactory overall distribution but also as nearly a zero pressure gradient as possible around the element. The importance of this procedure is evident from the fact that the presence of pressure disturbances has a four-way influence on the skin-friction values measured by the floating element, through:

(1) Changes upstream in the boundary layer

(2) Direct local effect of pressure gradient on the velocity profile

(3) Pressure forces acting on the sides of the floating element

(4) Disturbances of velocity profile caused by air flow around the floating element due to different static pressures in the gaps

Some of these effects, especially the pressure forces described in (3), can easily create errors of the same order of magnitude as the shear force itself. However, it is clear that the total error will vary from case to case. Hence, it would be extremely difficult to establish the values of the proper corrections, both because of the lack of knowledge of the phenomena involved and because of the difficulties associated with the accurate measurements of the pressure gradient itself. Therefore, instead of trying to determine the corrections theoretically or experimentally, it was decided to minimize the pressure force effect by designing the gaps so that they expand inward. It was also decided to make several measurements at approximately the same Mach number, introducing slight changes in the pressure distribution by varying the shape of the flexible nozzle. This technique was applied to the supersonic measurements and the results are discussed in detail in the section entitled "Measurements in Supersonic Flow." Although the measurements necessarily show some scatter, it is believed that their average values are relatively good indications of the shear stresses which would be obtained under ideal pressure distributions on the flat plate.

The Mach number given for each flow configuration was based on static pressure measured at the element.

Effect of Gaps Around Floating Element

In the interpretation of the results it was realized that the previous practice of using the actual area of the floating element in computing the shear stress from the measured force was not quite correct. It is very unlikely that introduction of a spanwise gap into a smooth surface would decrease its total drag. Therefore, it can be reasonably expected that a force equal to at least the skin friction based on the gap area acts on the solid surfaces as pressure forces on the sides of the gap and as increased skin friction on the surfaces exposed to flow.

Assuming that the gaps upstream and downstream of the floating element are approximately equal, the above reasoning leads directly to the conclusion that the proper reference area of the element is at least its actual area plus one-half of the surrounding gap area. The value of the "gap drag" in each case must naturally depend on the relative size of the gap as compared with a local characteristic length of the boundary layer.

Unfortunately, no theoretical or experimental investigations of the gap drag were found in the range of interest for the present case. Some measurements with large gaps, of the order of one-half or more of the boundary-layer thickness, have been reported in references 6 and 7 (both quoted in the more readily available ref. 8). The results indicate an effect of approximately twice the magnitude of the skin friction based on the gap area. Applied to the problem on hand this means that the total gap area should be added to the actual element area in order to obtain correct values of shear stress. It is believed, however, that the effect of the relatively very much smaller gaps of the present case, with the flow mechanism more dominated by viscosity, is likely to be smaller than that of the large gaps.

It was therefore judged to be reasonable to choose a reference area between the two values discussed above. In view of the lack of clearly conclusive information and of the fact that the gaps upstream and downstream are ordinarily not equal, thus permitting additional errors due to any asymmetry in the gap force distribution, it was further decided to attach a rather large range of systematic uncertainty to the present results. With a gap area of 11 percent of the floating-element area, the supposed minimum additive area is $5\frac{1}{2}$ percent and the maximum, 11 percent. The reference area was chosen halfway between these values and the uncertainty was estimated to be ±3 percent, with the range of uncertainty extending below the minimum and above the maximum.

It is apparent that more accurate information about the gap effect could be gained by an experimental investigation where different gap widths were employed for the same measurements, or different size gaps could be built into the element itself and the drag increase measured. It was felt, however, that in view of the limitations imposed by the small physical dimensions and by experimental scatter (discussed in the following sections) the present experimental setup would not be wholly satisfactory for this purpose. Such an investigation would best be performed with a larger instrument and in a wind tunnel with extremely uniform and easily controllable flow conditions.

From the practical point of view it is emphasized that in case of larger floating elements there is no need to increase the gap size beyond that required by resolution of the displacement transducer and feasibility of cleaning. Thus, in large instruments, and especially in those with null-type transducer systems, the gap effect can easily be made, if not entirely negligible, at least much smaller than that in the present case.

Accuracy of Skin-Friction Measurements

With ±1 percent repeatability of calibration and ±3 percent probable magnitude of the gap effect, measurements of the surface shear stress, performed in a zero pressure gradient, can be obtained with a maximum error of ±5 percent. In actual experiments, and in their evaluation, additional errors are introduced into the skin-friction coefficients by inaccuracies in the determination of flow parameters (dynamic pressure and Reynolds number) and by pressure disturbances in the free stream. These errors depend on the individual flow configurations and are partly reflected in the scatter of the measured values. The principal systematic uncertainty in this group is connected with the definition of proper reference Reynolds numbers and is discussed in detail in the next section.

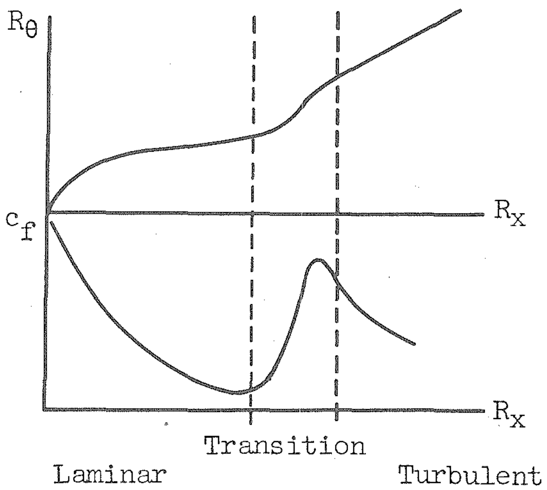
REMARKS ON TURBULENT BOUNDARY LAYERS

AT LOW REYNOLDS NUMBERS

The results of theoretical and experimental investigations of the compressibility effect on turbulent skin friction have been generally expressed in the form of the ratio of compressible to incompressible skin-friction coefficients for identical streamwise Reynolds numbers. This procedure has been satisfactory at the high Reynolds numbers of most of the previous experimental data. However, some doubts have arisen regarding the validity of the method for interpretation of the current experiments, which were performed at relatively low Reynolds numbers. The difficulty in question is a direct consequence of a certain arbitrariness in the definition of the Reynolds number as a characteristic quantity of the turbulent boundary layer. It is the purpose of this section to discuss the problem in detail and to propose suitable procedures for analyzing the results of the present investigation.

Transition From Laminar to Turbulent Flow

If a smooth flat plate large enough to develop transition into turbulent flow is considered the general picture is as follows:



$$c_f = \frac{\tau_w}{q} = 2\,\frac{dR_\theta}{dR_x}$$

$$R_\theta = \int_0^\infty \frac{\rho u}{\rho_\infty U}\left(1 - \frac{u}{U}\right)dR_y$$

$$= \frac{1}{2}\int_0^{R_x} c_f\ dR_x$$

$$R_y = \frac{\rho_\infty U y}{\mu_\infty}$$

It has been verified experimentally that laminar flow is well described by the existing theories. Very little is known about transition quantitatively, and it has been demonstrated that the characteristics differ from case to case. While it would be reasonable to expect that (at a certain Mach number) transition on flat plates of sufficient smoothness would occur for the same values of R_x (based on distance from the leading edge) and would bear a certain relation to R_θ and c_f, it should be noted that, under actual physical conditions, disturbing effects always occur and in many cases are even deliberately introduced, often in a crude manner, to induce transition. Thus, in reality, the position and structure of transition do not follow any specified law; consequently, a well-defined starting point for the turbulent boundary layer in terms of R_x, R_θ, or c_f cannot be expected.

Fully Developed Turbulent Boundary Layer

Downstream of the transition region, a relation between c_f and R_θ has been experimentally shown to exist for incompressible boundary layers (there is an extensive discussion of available data in ref. 9). Whether this relation is truly independent of secondary effects, such as the free-stream turbulence level, has not been definitely shown; however, it is believed to be sufficiently well founded to be used under ordinary conditions. Further, there appears to be no evidence against an extension to compressible flows, although the experimental support cannot yet be considered sufficient. In the following discussion it is assumed that, downstream of the transition region on a flat plate, a relation of sufficient practical accuracy exists connecting c_f, R_θ, and M. Where this relation holds the turbulent boundary layer is described as "fully developed."

Reynolds Number of a Turbulent Boundary Layer

Since $c_f = 2(dR_\theta/dR_x)$, the relation assumed above connects R_x with c_f, R_θ, and M, except for an arbitrary constant which requires the introduction of one further condition. Although not always explicitly pointed out, the same situation has also occurred in theoretical considerations. In Von Kármán's original incompressible results (refs. 10 and 11) and Coles' reevaluation of them (ref. 9), for example, a condition that $R_x = 0$ when $c_f = \infty$ has been applied, although the "point of origin" thus defined does not have any counterpart in a physical configuration. In experimental work, the data have either been fitted to an assumed theoretical law (refs. 12 and 13) or based on a length such as that from the leading edge (refs. 1, 2, and 3 and a large number of other investigations), tripping device, or shear-stress peak in the transition region (ref. 9). These methods of defining point of origin need not be such as to permit a comparison of results in a unique manner. At least for the present, there seems to be no clear way out of this difficulty.

It is fortunate that the combined laminar and transition regions seem to be limited in size to a Reynolds number from the leading edge of the order of 10^6 (at least up to moderate supersonic Mach numbers). Thus, by working at sufficiently high Reynolds numbers a large part of the distance between the station of measurement and the leading edge is covered by the fully developed turbulent boundary layer. With increasing Reynolds number, one can then expect to be able to establish an asymptotically improving description of the boundary-layer characteristics in terms of a Reynolds number based on any of the physical lengths mentioned above. The contribution of the laminar and transition regions to R_θ and R_x becomes so small relatively that differences in position of transition cannot influence the results to a significant extent. That this kind of behavior is actually true has been demonstrated by a large number of experiments with widely varying conditions near the leading edge. (Graphical summaries are given in, e.g., refs. 14 and 15.) As a practical procedure one could use the distance from the leading edge of a flat plate as the basis for Reynolds number whenever the ratio of the distances of the point of measurement and the end of the transition region from the leading edge is large. Thus, for incompressible flow with a natural transition region somewhere near $R_x = 10^6$ (from the leading edge), one could expect the measurements to be in reasonable agreement at $R_x > 10^7$. This is, in fact, what has been observed.

The fact that some experiments with artificial tripping (e.g., ref. 1), where the Reynolds number has been based on the distance from the leading edge or tripping device, follow Von Kármán's law closely down to about 2×10^5 is merely a reflection of the fact that in terms of Von Kármán's law the origin $R_x = 0$ of the fully developed layer falls close to the normally occurring locations of the transition zone. It should be emphasized, however, that in a rigorous sense the Reynolds number used in the description of these experiments has little significance. The only correct way of identifying a fully developed turbulent boundary layer in terms of a suitably defined streamwise Reynolds number is by means of the local value of c_f or R_θ.

For the compressible turbulent boundary layer, there is, at present, no theoretical relationship comparable with Von Kármán's incompressible law. Several attempts have been made in this direction (refs. 12, 16, 17, 18, and 19; see summary in ref. 9) but none of them has the support of sufficient experimental evidence. Furthermore, unless a theoretical law were a direct extension of Von Kármán's incompressible law, it obviously would not need to have any connection with the definition of Reynolds number in that case. Many of the relations referred to above are, in fact, attempts at such an extension, but their theoretical foundations differ from case to case and are often artificial. It is believed that the present stage of knowledge does not allow a meaningful definition of Reynolds number to be established, except at the larger Reynolds numbers where the leading edge, tripping device, or some characteristic of transition can again be used as a reasonable basis in the same manner as in the incompressible case. That the compressible fully developed turbulent boundary layer actually follows a similar asymptotic relation has been demonstrated by Coles (refs. 4 and 9). At the low Reynolds numbers it is evident, however, that in a rigorous sense all one can hope to obtain from an experimental investigation of a compressible turbulent boundary layer is the supposed relationship between R_θ, c_f, and M.

Comparison of Compressible and Incompressible Boundary Layers at Low Reynolds Numbers

In order to attach an approximate streamwise Reynolds number to measurements performed in compressible flow in the low Reynolds number range one would, for instance, be tempted to assume a point of origin close to the transition zone, as in the incompressible case. Despite the fact that in compressible flow both the laminar and the transition regions are different from incompressible flow (as regards both the streamwise extent, in terms of Reynolds number, and the structure of the regions, in terms of R_θ and c_f), Coles has found the use of peak shear, as an origin for Reynolds number, to give a reasonably good description of his experimental results. The application of the peak-shear criterion to the results of the present investigation is discussed in the section entitled "Measurements in Supersonic Flow."

An apparent way out of the difficulty would be to compare the skin-friction coefficients, not at the same values of R_x but at the same values of R_θ. This would leave no arbitrariness in the evaluation of experimental results. At low values of R_θ, however, it may not always be possible to make such a comparison because the minimum R_θ at the beginning of the fully developed region probably depends on Mach number. Also, the fact that momentum thickness can be measured only indirectly by means of a velocity survey across the boundary layer presents a disadvantage.

The highest Reynolds numbers (measured from the leading edge) occurring in the present experiments were about 1.2×10^6. All data thus fall in the range where the streamwise Reynolds number cannot be defined accurately. In view of the above discussion it was decided to use the following methods in the analysis:

(1) Subsonic measurements: Reynolds number based on distance between leading edge (or trip) and the point of measurement

(2) Supersonic measurements: (a) Reynolds number based on distance between observed shear peak in transition region and point of measurement and (b) Reynolds number based on momentum thickness computed from measured velocity profile

Since theoretical or experimental incompressible results based on the same criteria are not available, except in the case of momentum thickness, the relations of Von Kármán (ref. 11) and of Coles (ref. 9) are used instead.

MEASUREMENTS IN SUBSONIC FLOW

Although the main aim of the present experiments was to investigate turbulent skin friction at the supersonic end of the transonic regime, it was decided to obtain a few subsonic measurements in order to compare them with existing results. Several Mach numbers, beginning at 0.18, were covered; the Reynolds numbers ranged from 0.33×10^6 to 1.20×10^6.

Experimental Technique

In the subsonic experiments the instrument was placed directly above the flexible nozzle, which was used to great advantage in adjusting the pressure gradient on the flat plate. The pressure distributions are shown in figure 6. Although the measurement at $M = 0.97$ is not satisfactory because of a supersonic zone at the leading edge, it is included as an illustration of conditions obtainable close to sonic speed.

At subsonic speeds the expansion of the flow around the sharp, half-wedge, leading edge has some destabilizing effect on the boundary layer. However, it was concluded from preliminary experiments that this effect alone was not sufficient to create a satisfactory turbulent flow on the flat plate. Several additional tripping devices were tried until, finally, a three-dimensional rake cemented on the leading edge was adopted. The final trip consisted of 0.02-centimeter-diameter pins about 0.3 centimeter long mounted 0.3 centimeter apart and swept back about 45°. Variations of this rake, involving slightly different pin dimensions and mountings, had no additional influence on the local skin friction measured at the element. Figure 7 shows the effects of various tripping devices.

The Reynolds number used in the evaluation of these measurements was based on the distance between the leading edge (or trip) and the measuring element, as was done in Dhawan's experiments. A closer investigation of the transition region was not attempted in the subsonic measurements.

Discussion of Results

The measured local skin-friction coefficients are presented in table I and figure 8. Incompressible reference curves are the original Von Kármán - Kempf result (ref. 11), a proposed modification by Dhawan, and the most recent reevaluation of available data by Coles.

The results show a scatter of approximately ±5 percent. This scatter is believed to be largely due to the effect of free-stream pressure disturbances. Referring to the estimated ±5-percent uncertainty of the shear-stress measurement and the indeterminacy of the Reynolds number, it is not considered worth while to make a more detailed analysis of the accuracy of the subsonic experiments. It can be concluded, however, that the measurements are in substantial agreement with previous evidence, especially with Dhawan's work, and indicate that compressibility slightly decreases the skin friction at the higher subsonic Mach numbers.

MEASUREMENTS IN SUPERSONIC FLOW

The supersonic experiments were performed at two Mach numbers, about 1.50 and 1.75. The Reynolds number, based on the distance between leading edge and measuring element, was approximately 10^6 in both cases.

Previous experimental information on turbulent skin friction at these speeds is not very extensive, and on local skin-friction values it is extremely scarce. No direct force measurements, other than Dhawan's preliminary ones, have been reported for flat plates. Other results could be obtained by differentiation of momentum-thickness distributions reported in several references, but this procedure is very inaccurate.

At Mach numbers above 2, Coles (refs. 4 and 9) has recently obtained extensive measurements of local skin friction by the floating-element technique. Cope (ref. 20) has reported some local measurements at $M = 2.5$ by means of a Stanton tube, but the use of calibration obtained in low-speed flow renders his results doubtful.

In the absence of adequate theoretical work, the present experiments can be compared with the behavior interpolated from existing incompressible data and Coles' measurements at higher Mach numbers.

Experimental Technique

In the supersonic experiments, the instrument was placed in the test rhombus of the wind tunnel. Mach number and pressure distribution on the flat plate were controlled by the flexible nozzle mechanism. Because of the large streamwise extent of the model and the limited amount of control of the nozzle shape, it proved exceedingly laborious to establish satisfactory flow conditions, a difficulty experienced by Dhawan in his preliminary measurements. However, with the greater variety of installations allowed by the present design, useful configurations were found at Mach numbers around 1.50 and 1.75. At lower supersonic Mach numbers, a detached shock wave from the housing of the force-sensing mechanism interfered with the flow ahead of the leading edge and no measurements were attempted in this range. Several measurements were taken at each Mach number with slight differences in pressure distributions. Typical pressure distributions are presented in figure 9.

Since the Reynolds number per unit length available in the GALCIT transonic wind tunnel is about the same for high-subsonic and supersonic speeds, measurements were first made using the same trip rake as in the subsonic experiments described in the preceding section. These measurements yielded values so much below those expected that a closer investigation of the transition region was deemed necessary. For this purpose a variable trip based on the principle of air injection described in reference 21 was devised. This trip consisted of a row of 0.05-centimeter-diameter holes close to the leading edge through which air jets were directed into the boundary layer. The mass flow of the trip air was measured by means of a Fischer and Porter floating-ball-type flowmeter. The modified flat plate is illustrated in figure 2.

For an investigation of the transition region, a slitlike total-head tube was employed which could be moved along the flat plate surface within the limitations of the traversing mechanism. Although its opening was only about 0.02 centimeter wide, it was too large to be used as a Stanton tube for quantitative measurements of shear. However, it is believed that the probe still provided a qualitative indication of the actual surface shear stress.

Some velocity-profile measurements were made using another slitlike total-head probe similar to the one used in the transition investigation. The probe position was observed by means of a cathetometer.

Effect of Variable Tripping on Transition and Local Skin Friction

Surface total-head surveys of the supersonic configurations are presented in figures 10 and 11. The vertical coordinate $\sqrt{\frac{p_T - p}{q}}$ was chosen because it would be proportional to the local skin-friction coefficient if the probe indicated the average dynamic pressure on its face due to a uniform velocity gradient.

The parameter varied to obtain the curves in figures 10 and 11 is the ratio of trip-air mass flow per unit width of the plate to the mass-flow defect $\rho_\infty U\delta^*$ of the laminar boundary layer at the location of the trip. Computation of the mass-flow defect was based on reference 22, where the following formula is given for the displacement thickness of a compressible laminar boundary layer:

$$\delta^* = \delta^*_i\left[1 + \frac{\gamma - 1}{2}\left(1 + \frac{\theta_i}{\delta^*_i}\right)M^2\right]$$

Figures 12 and 13 present typical variations of the skin friction measured at the floating element when the trip mass flow was varied. At both Mach numbers the skin friction decreased with increasing mass flow from a peak value (when the peak shear of the transition zone passed over the element) to a small shelflike constant region followed by further slow decrease. The skin-friction measurements were taken normally in the shelflike region. Some values based on the slow decrease region were obtained at $M = 1.75$, although it is possible that the corresponding location of peak shear was influenced by local effects due to air jets. Some experiments, carried out with a trip-hole spacing half that described in figure 2, did not show any change in the transition characteristics.

It should be noted that the natural transition at $M = 1.75$ was just beginning at the element, corresponding to a Reynolds number of 10^6, whereas at $M = 1.5$ laminar flow could not be maintained beyond about $R_x = 3 \times 10^5$. This corroborates other experimental evidence of the stabilizing effect of compressibility which has been reported by several investigators (a recent survey of the problem has been presented in ref. 23). A related phenomenon may be the fact that the distances between the trip jets and the shear peaks, for a given trip-air mass flow, were noticeably larger at $M = 1.75$ than at $M = 1.5$.

The behavior of transition shown in the figures referred to in this section was not quite repeatable, probably owing to variable deposits of dust on the plate. The results are, therefore, not to be considered directly transferable to other experimental configurations.

Discussion of Results in Terms of Streamwise Reynolds Number

The measurements of supersonic skin friction in terms of Reynolds number based on distance from location of peak shear are presented in table II and figure 14.

The measured points were used to obtain values of c_f/c_{f_i} at one Mach number in each range by extrapolating from each point, using the slope of Von Kármán's theoretical curve (ref. 16). These values were then averaged. In view of the small shifts involved, this approximation is believed to introduce negligible error. The scatter before averaging was ±3 percent. Since the distribution was clearly non-Gaussian, error theory could not properly be applied and the probable error of less than ±1 percent which it gives is no doubt too small.

The scatter is thought to be due to disturbances in the free-stream pressure field and difficulties in locating the peak shear, rather than to errors in the measurement of shear stress itself. It should therefore be considered separately from the more or less systematic shear-stress-measurement error of ±5 percent in the measurements under ideal conditions. The additional uncertainty arising from the indeterminacy of the compressible and incompressible Reynolds numbers is connected with the validity of the whole procedure of evaluation of the results, and a reliable estimate of these errors cannot be attempted. Therefore, a discussion of accuracy beyond that given for the shear-stress measurement and the scatter of the present results is believed to be pointless.

It should be noted that according to Von Kármán's estimate (ref. 16) the value of c_f/c_{f_i} at a certain Mach number should decrease slightly with increasing Reynolds number. In view of this possibility the average results in terms of streamwise Reynolds number indicate a good agreement with behavior interpolated from incompressible results and those of Coles at higher Mach number.

Discussion of Results in Terms of Momentum Thickness

The results obtained in terms of dimensionless momentum thickness are presented in table III and figure 15. Velocity profiles used in the evaluation of the momentum thickness are given in figures 16 and 17.

These results involve too few measurements to permit an accuracy estimate on the basis of scatter. The momentum thicknesses were determined by graphical integration from velocity profiles, which in turn were computed from the total-head readings applying the constant-energy assumption. Errors arise from this procedure as well as from inaccuracies of the cathetometer readings. No velocity-gradient correction has been made to the total-head-tube position in view of the recent experimental results obtained in supersonic turbulent boundary layers (refs. 13 and 24). Without attempting to estimate the total maximum error numerically, it is believed to be of the order of several percent.

The results indicate the decrease of skin friction of these Mach numbers to be slightly less than that obtained when streamwise Reynolds number is used. It should be noted that the $M = 1.75$ values seem high, although within experimental accuracy, compared with the $M = 1.5$ results. This may be an indication that the local effects of transition had not yet completely disappeared from the boundary layer at the element, which is quite possible in view of the lower Reynolds numbers obtained at this speed.

CONCLUDING REMARKS

The relatively small and compact GALCIT direct skin-friction meter developed by Dhawan has been further developed and applied to the investigation of high-speed turbulent boundary layers on a flat plate at Mach numbers up to 1.75.

Accuracy within ±5 percent can be obtained under ideal flow conditions. This is capable of substantial improvement in large instruments or with more information on the effects of gaps around the measuring element.

In the range of the present experiments, the lack of knowledge of the proper definition of an absolute streamwise Reynolds number for turbulent boundary layers introduces a fundamental difficulty into a comparison between compressible and incompressible skin friction at the same Reynolds number and necessitates the use of approximate methods for evaluation of the results. At Mach numbers of 1.5 and 1.75, a decrease of skin-friction coefficient from the incompressible value (based on Coles' relation) of approximately 10 percent was observed. A comparison of the measured skin-friction coefficients with incompressible values at the same dimensionless momentum thickness showed a slightly smaller compressibility effect.

California Institute of Technology,
Pasadena, Calif., July 6, 1953.

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TABLE I

SUBSONIC MEASUREMENTS WITH REYNOLDS NUMBER

BASED ON DISTANCE FROM TRIP DEVICE

M	R_x	c_f
0.18	0.33×10^6	417×10^{-5}
.20	.36	409
.26	.48	395
.27	.48	380
.31	.55	384
.33	.59	374
.37	.63	366
.46	.76	340
.49	.82	344
.56	.90	336
.57	.90	337
.65	1.00	330
.75	1.03	301
.85	1.12	301
[a].97	1.20	300

[a]At M = 0.97 a supersonic region existed near leading edge.

TABLE II

SUPERSONIC MEASUREMENTS WITH REYNOLDS NUMBER BASED ON DISTANCE FROM PEAK SHEAR IN TRANSITION REGION

M	R_x	c_f	Average results (a)
1.45	1.04×10^6	300×10^{-5}	M = 1.50 c_f/c_{f_i} = 0.89
1.48	1.04	294	
1.49	1.04	300	
1.50	1.02	291	
1.50	1.04	302	
1.50	1.02	302	
1.50	1.03	302	
1.50	1.03	292	
1.52	1.02	309	
1.52	1.02	307	
1.52	1.01	309	
1.52	1.01	307	
1.71	0.68	324	M = 1.75 c_f/c_{f_i} = 0.89
1.72	.67	321	
1.73	.66	321	
1.74	.67	323	
1.75	.67	323	
1.73	0.84	313	[b]M = 1.75 c_f/c_{f_i} = 0.90
1.74	.85	319	
1.74	.85	319	
1.76	.84	310	
1.76	.84	312	

[a]Values of c_{f_i} taken from ref. 9.
[b]Not plotted in fig. 16.

TABLE III

SUPERSONIC RESULTS WITH REYNOLDS NUMBER

BASED ON MOMENTUM THICKNESS

M	R_θ	c_f	Average c_f/c_{f_i} (a)
1.48	1,910 2,340	315×10^{-5} 302	0.92
1.52	1,610 2,030	324 307	0.91
1.75	1,780 1,920	323 319	0.93

[a]Values of c_{f_i} taken from ref. 9.

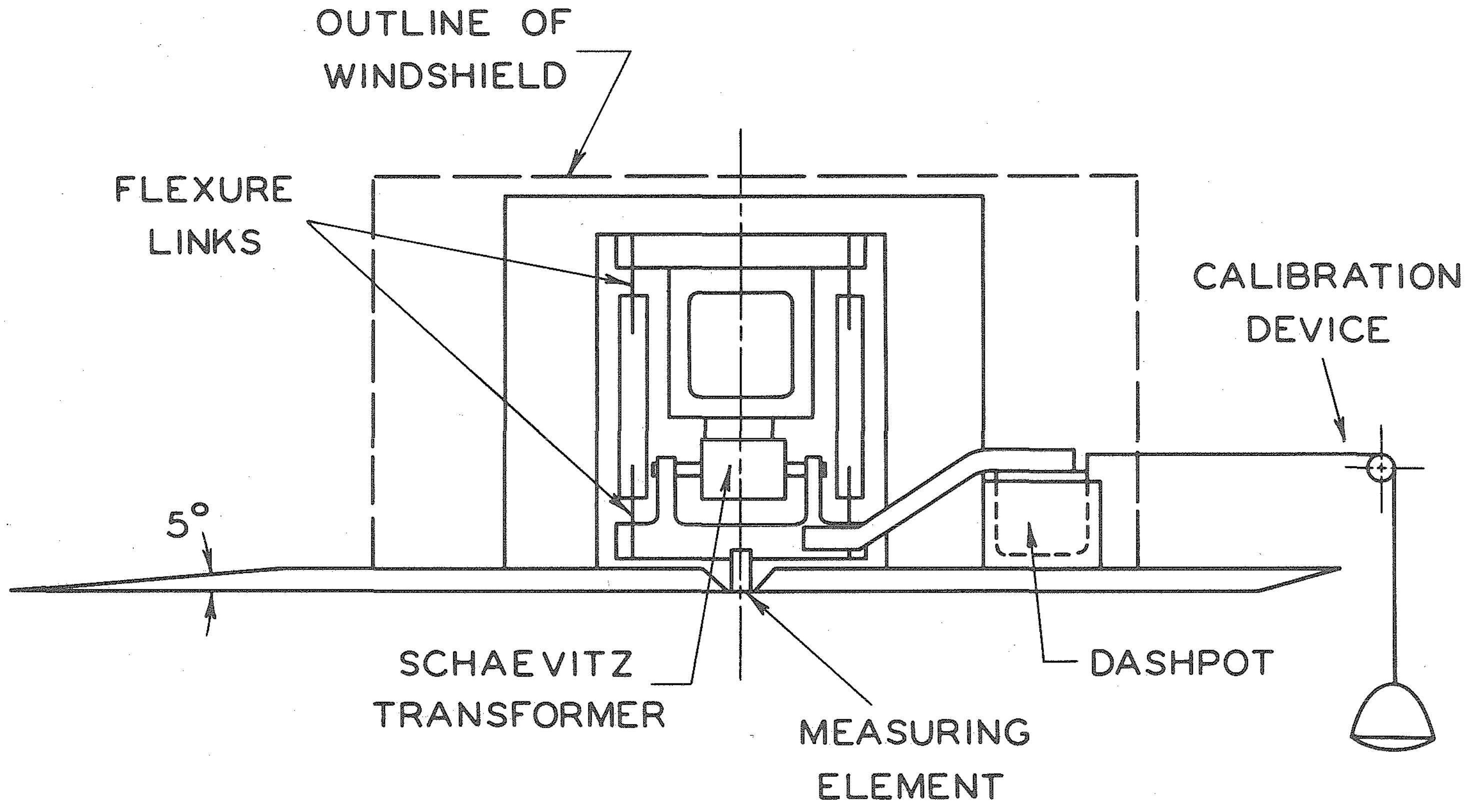

Figure 1.- Skin-friction meter.

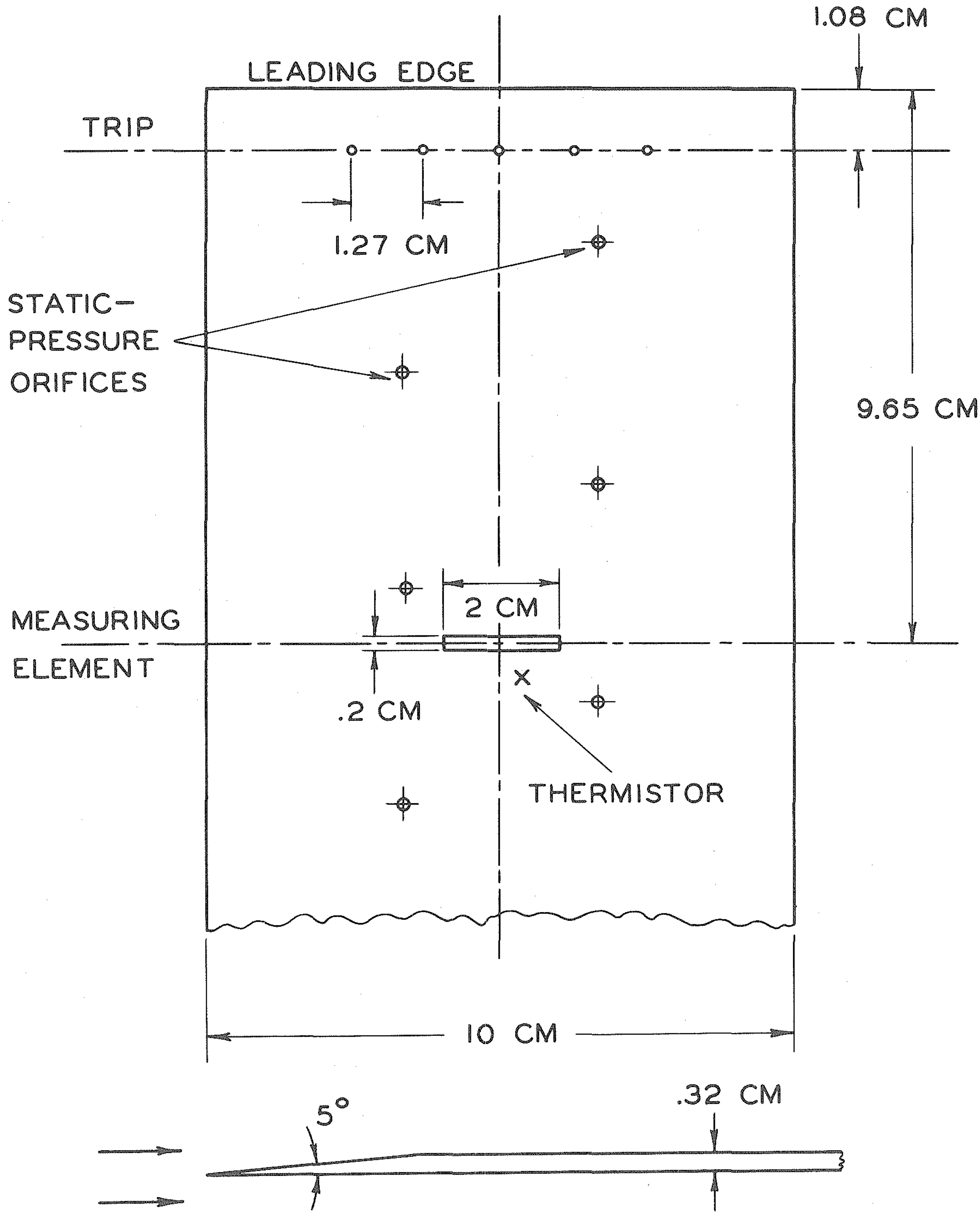

Figure 2.- Flat-plate configuration for supersonic experiments.

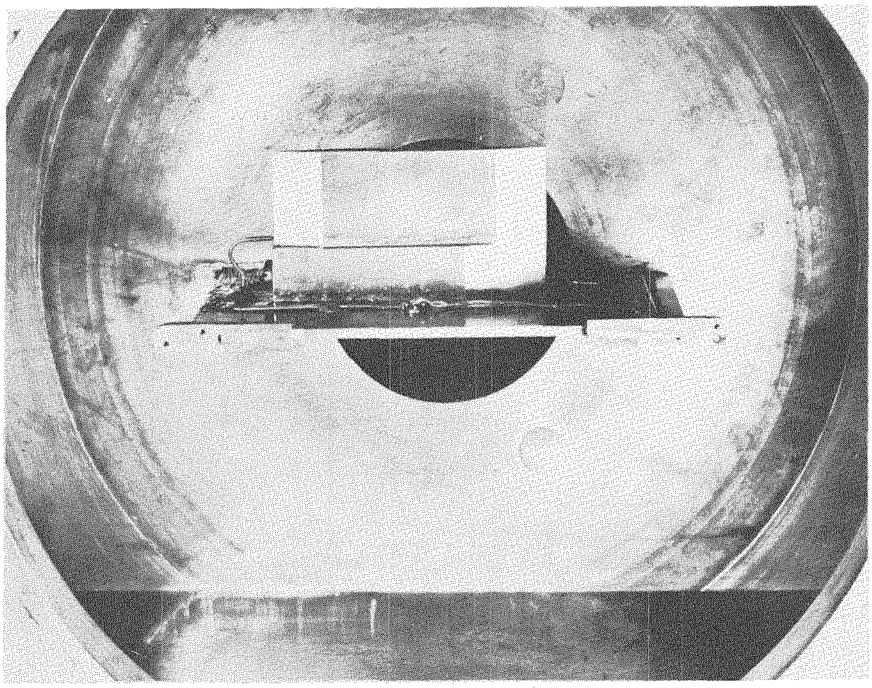

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Figure 3.- Installation of skin-friction meter in GALCIT 4- by 10-inch transonic wind tunnel.

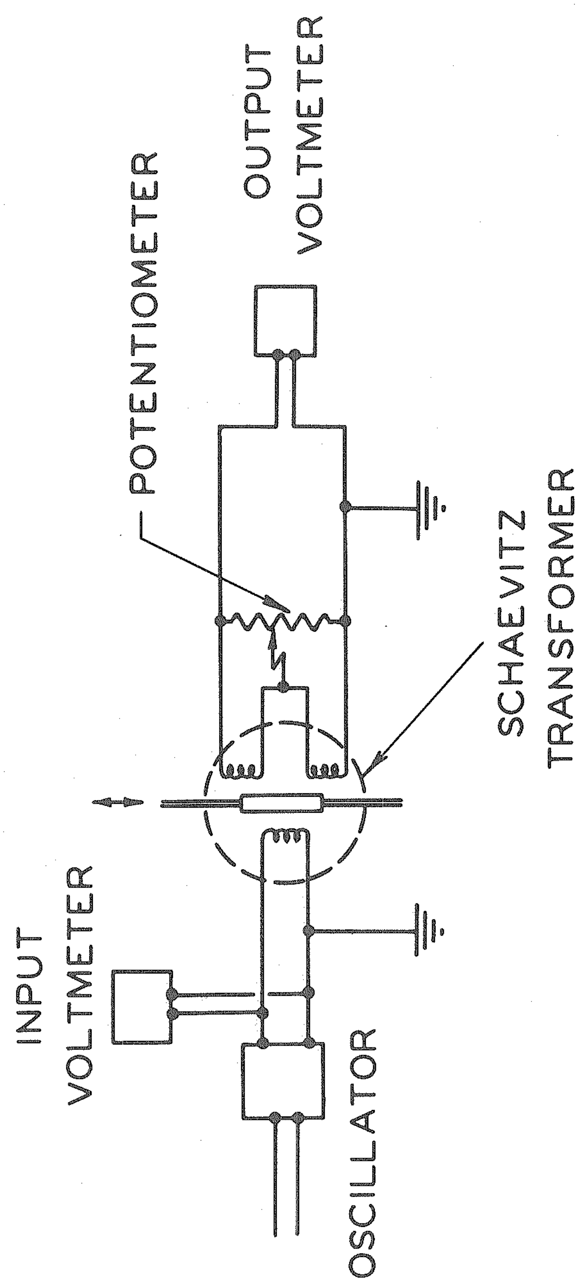

Figure 4.- Electrical circuit.

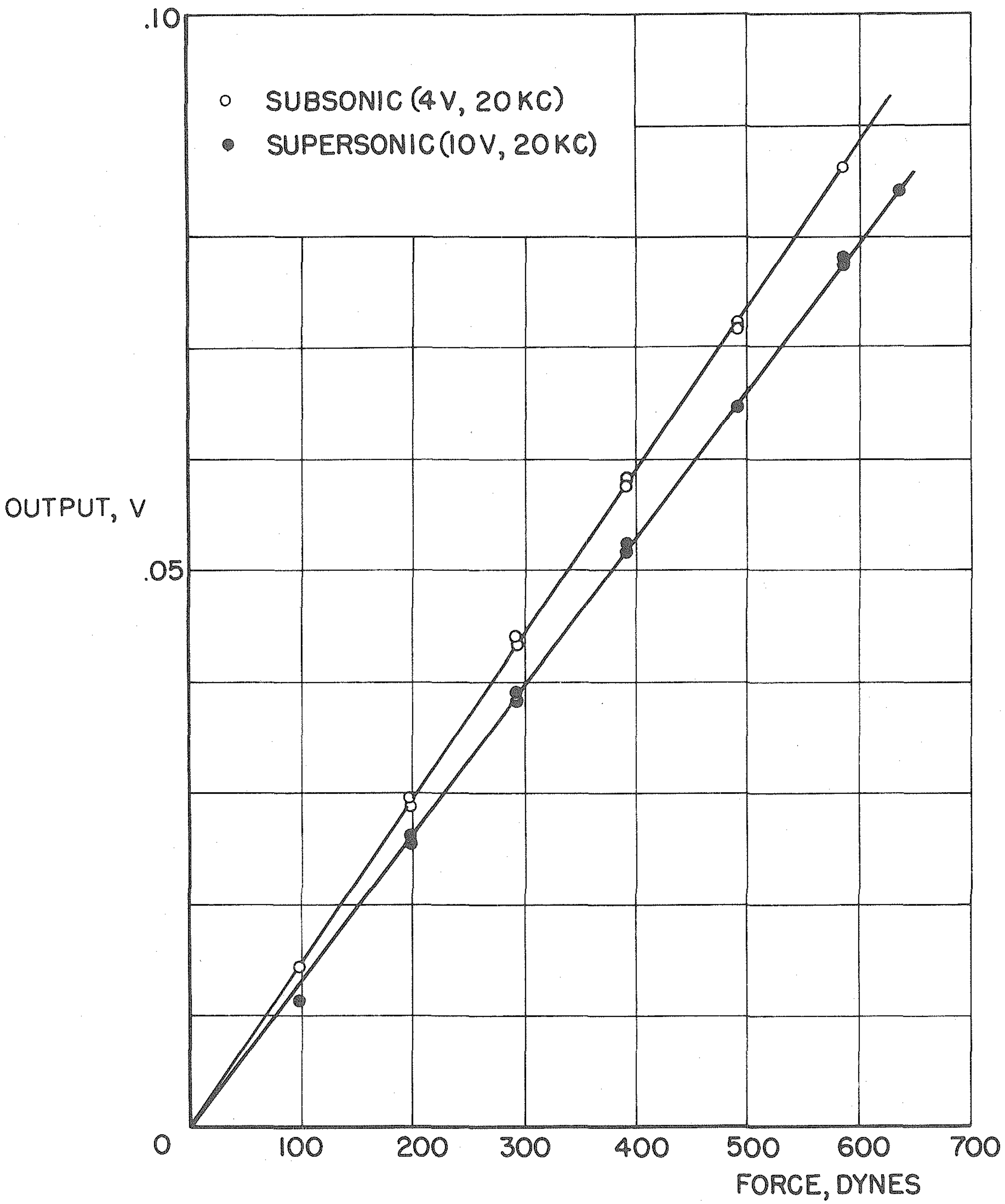

Figure 5.- Calibration of skin-friction meter.

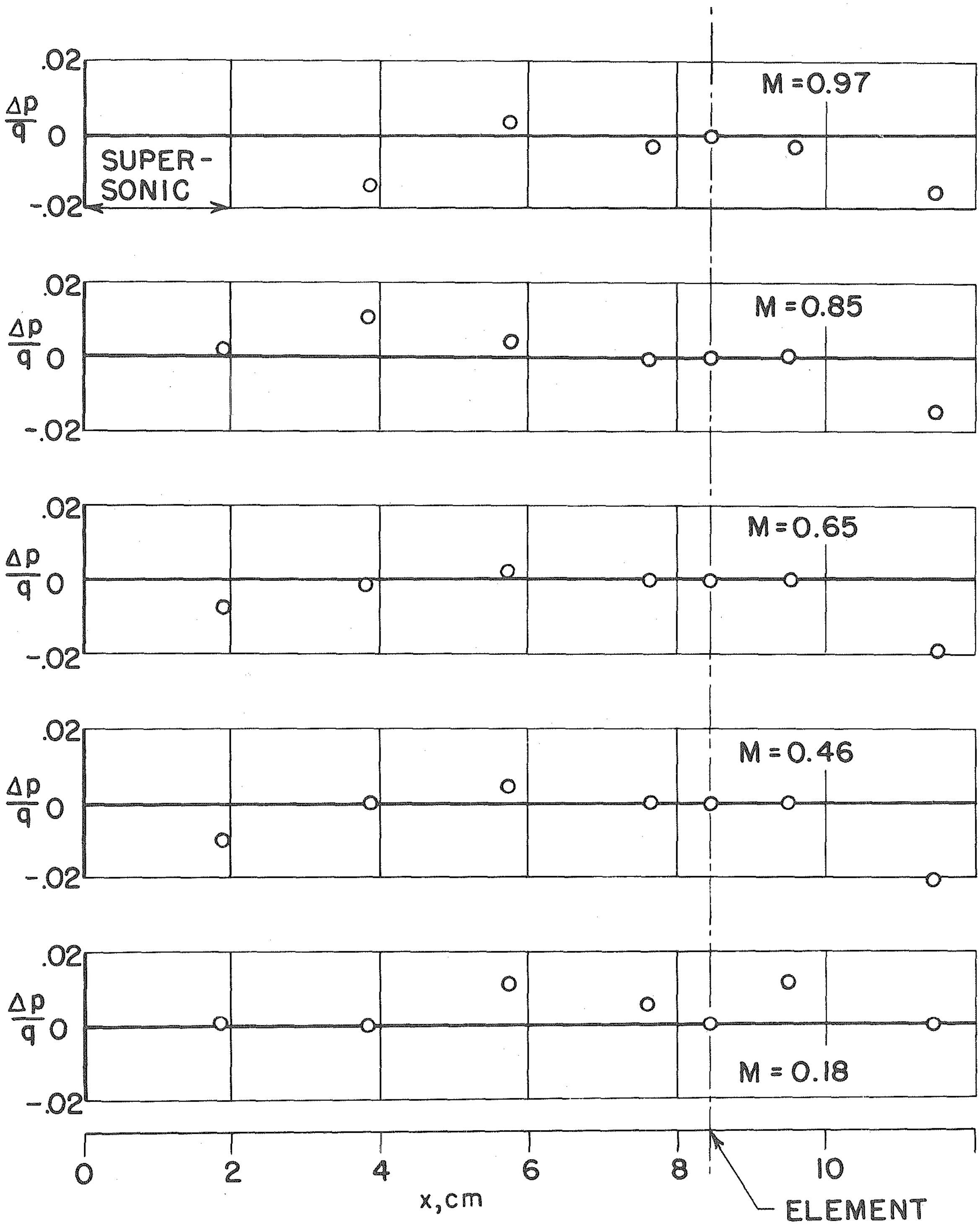

Figure 6.- Pressure distributions at subsonic speeds.

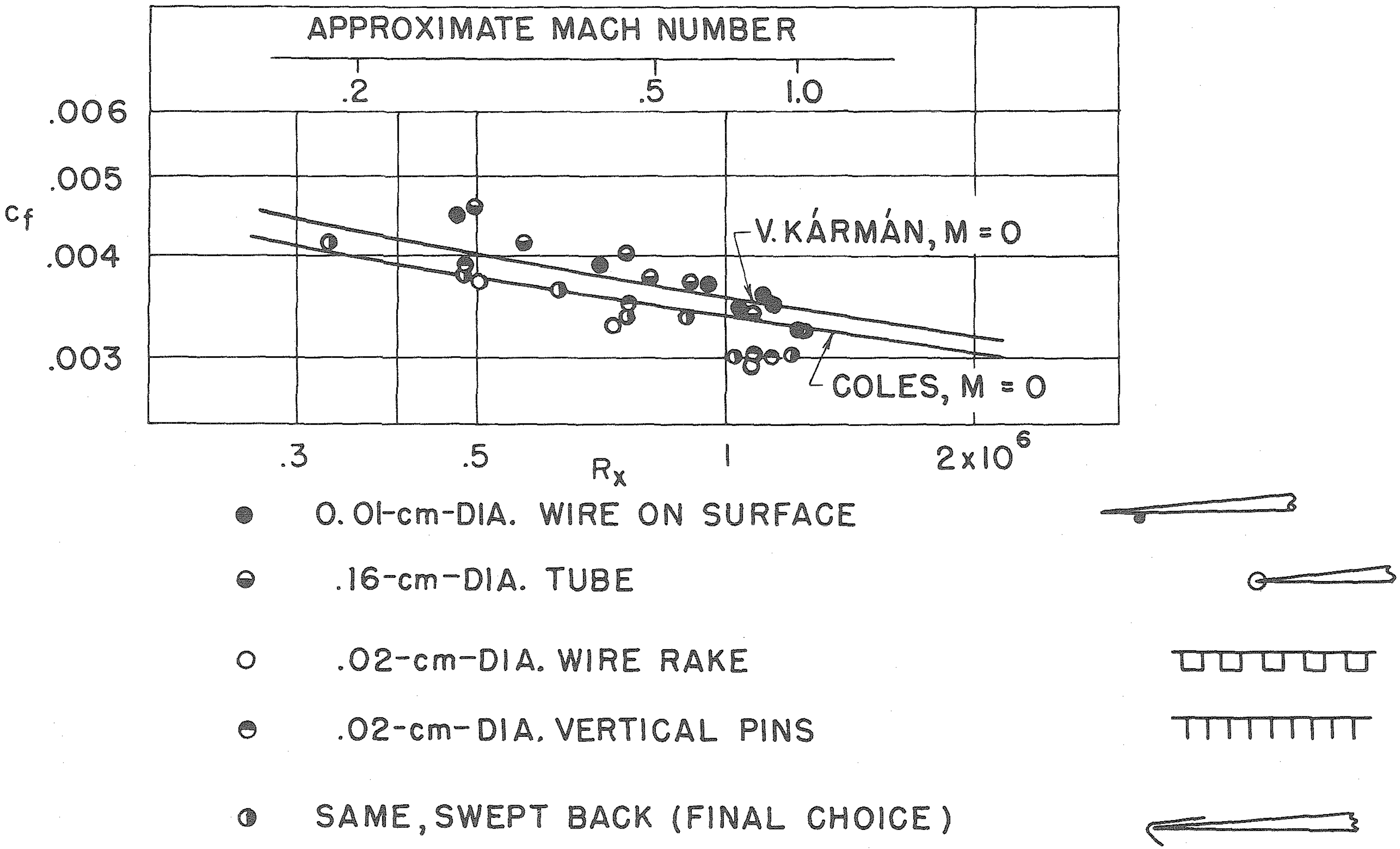

Figure 7.- Effect of method of tripping on local skin friction.

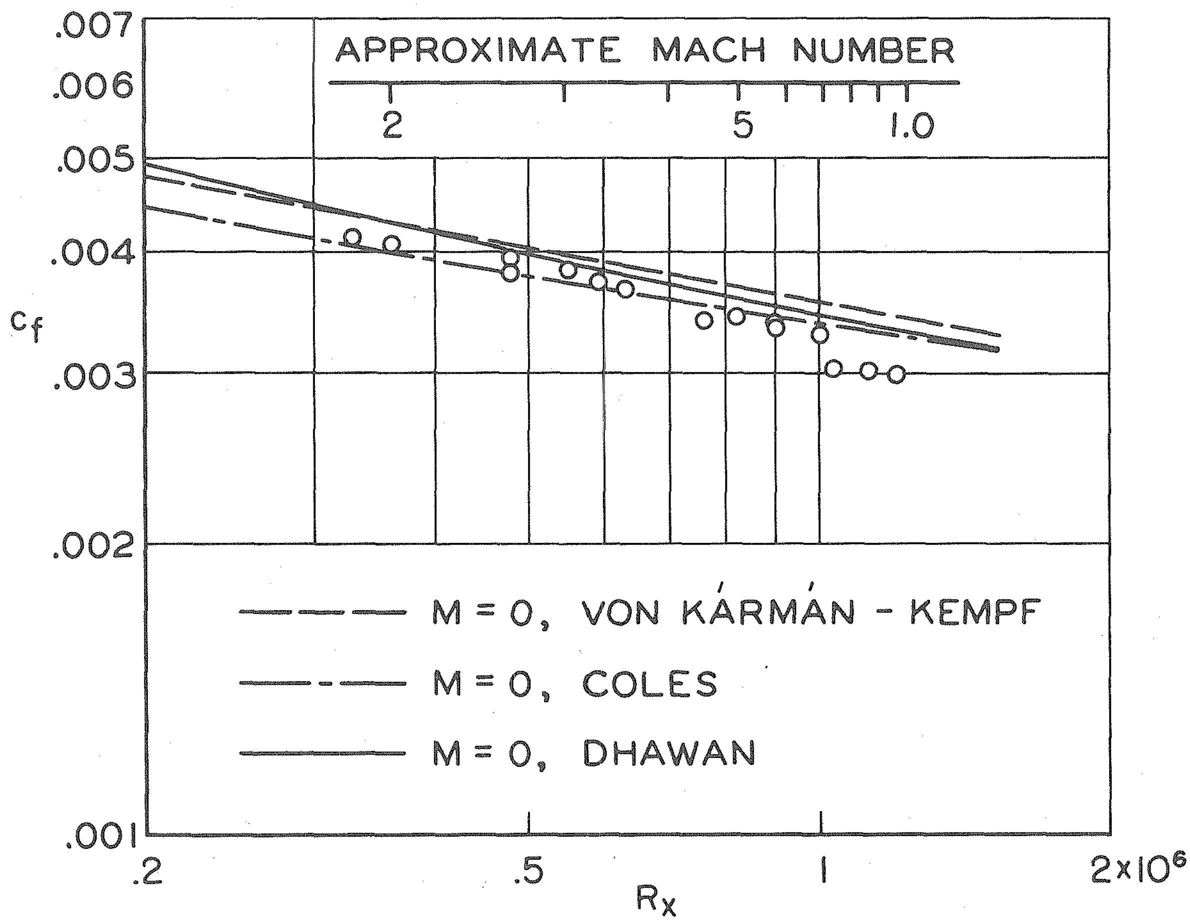

Figure 8.- Subsonic measurements.

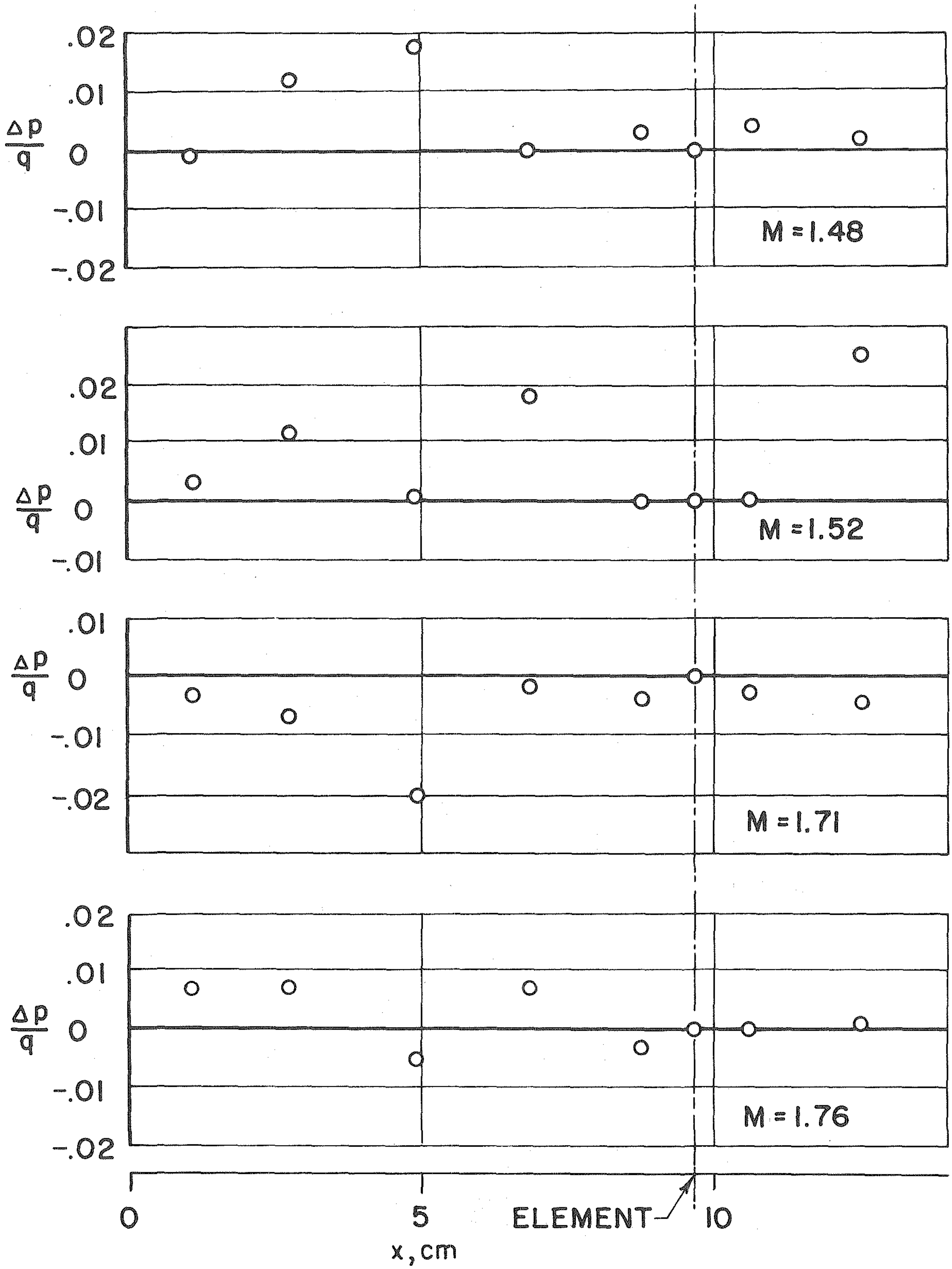

Figure 9.- Representative pressure distributions at supersonic speeds.

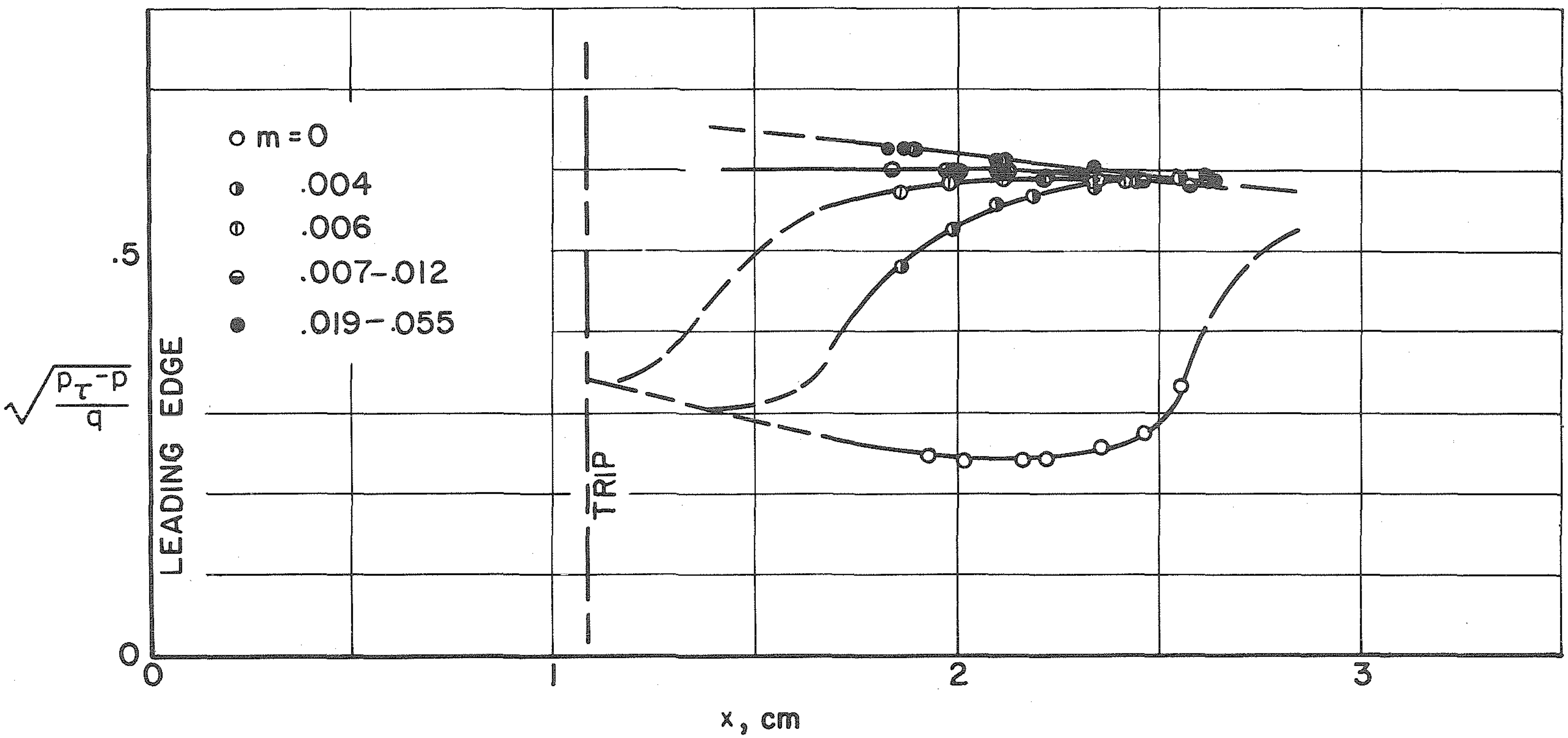

Figure 10.- Transition at $M = 1.48$ and $R = 1.34 \times 10^5$ per centimeter.

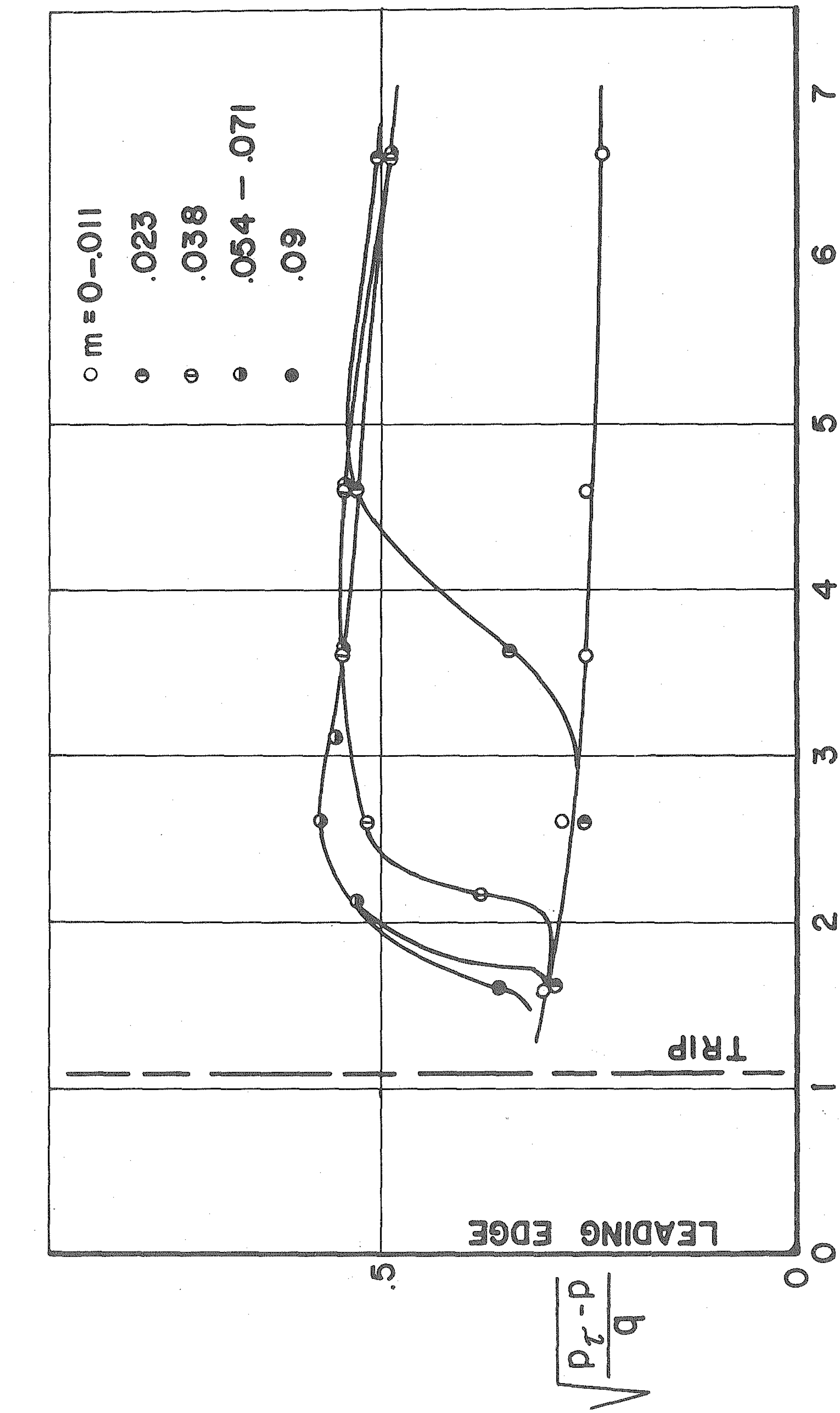

Figure 11.- Transition at M = 1.76 and R = 1.15 × 10^5 per centimeter.

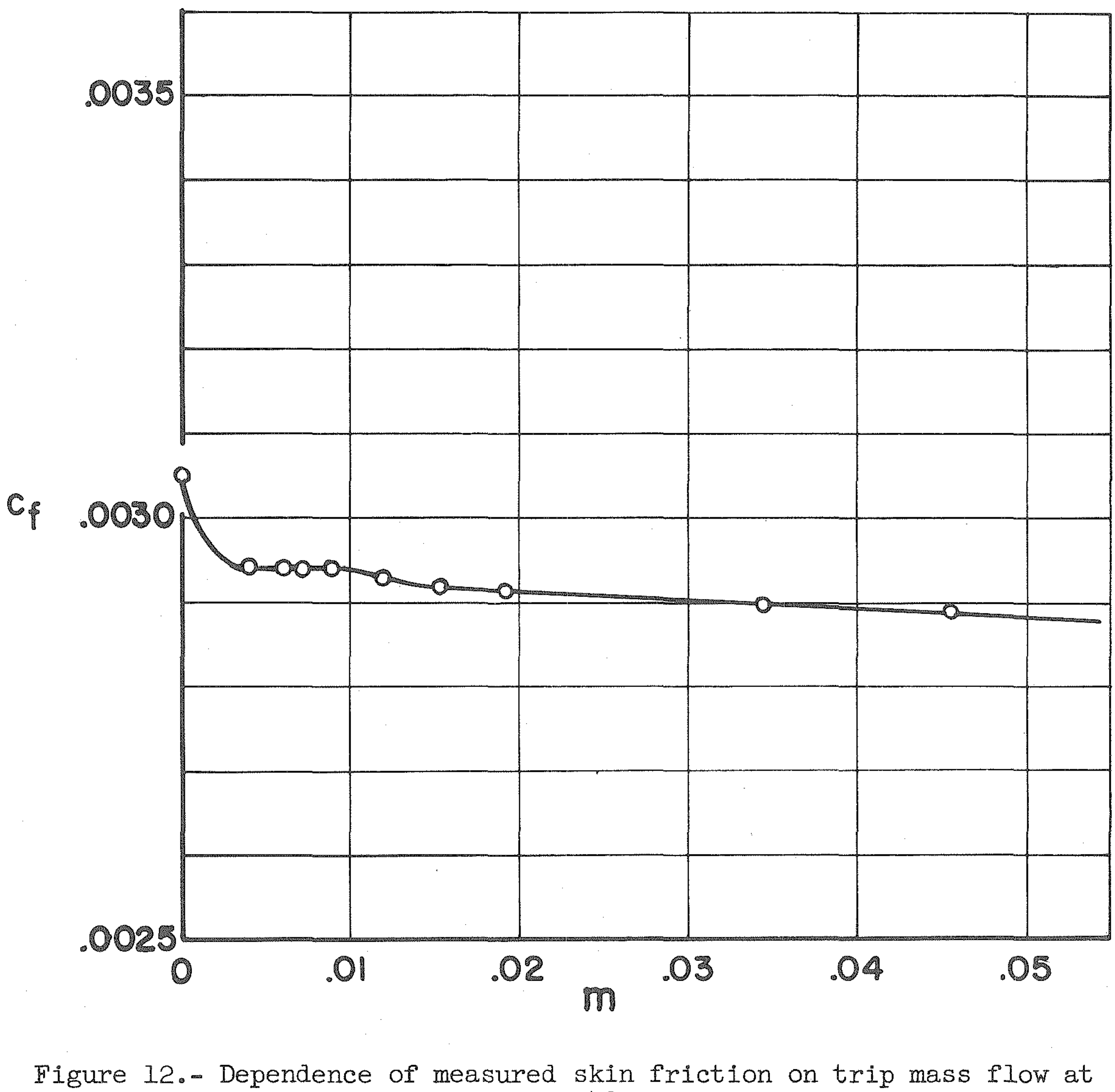

Figure 12.- Dependence of measured skin friction on trip mass flow at $M = 1.48$.

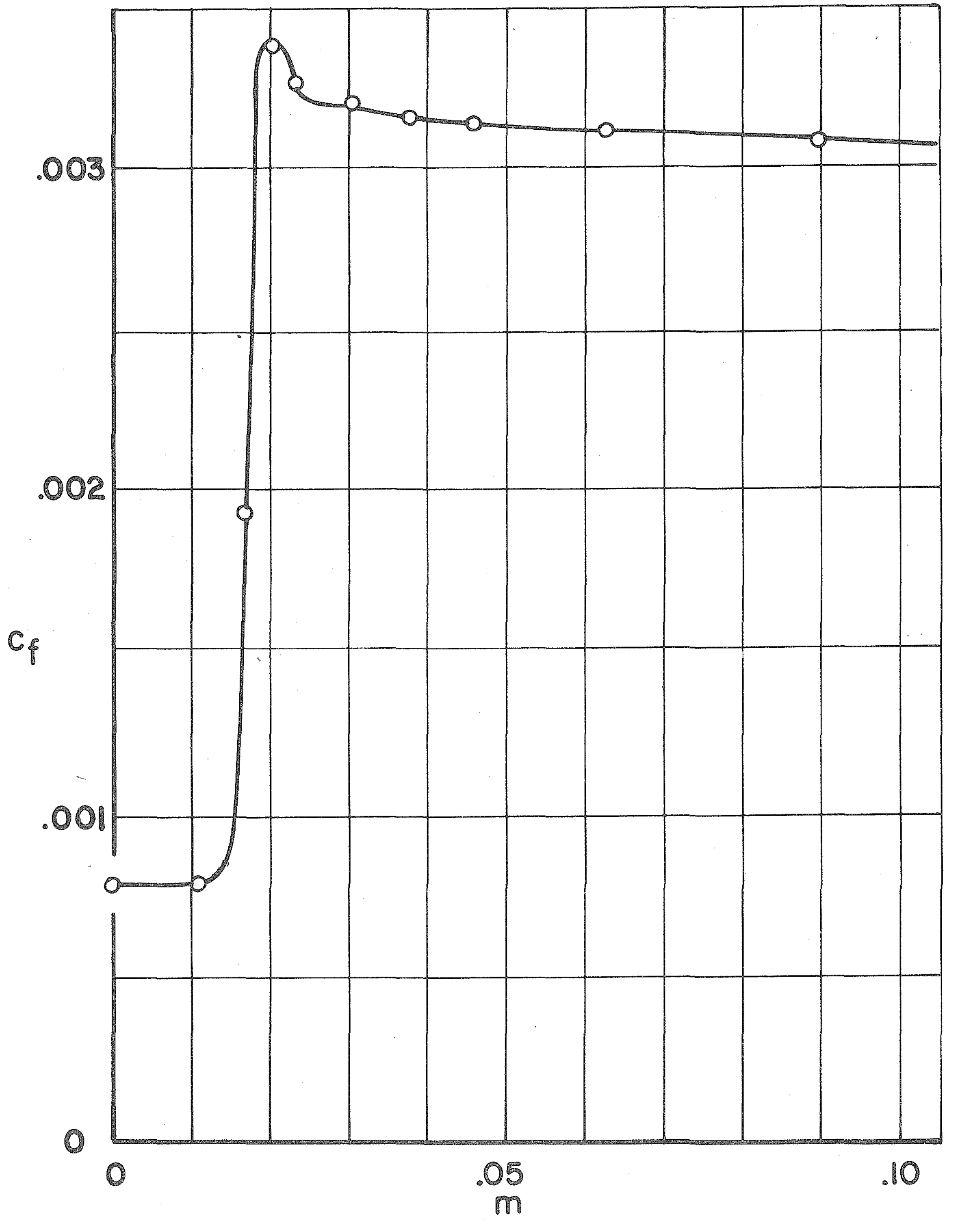

Figure 13.- Dependence of measured skin friction on trip mass flow at
M = 1.76.

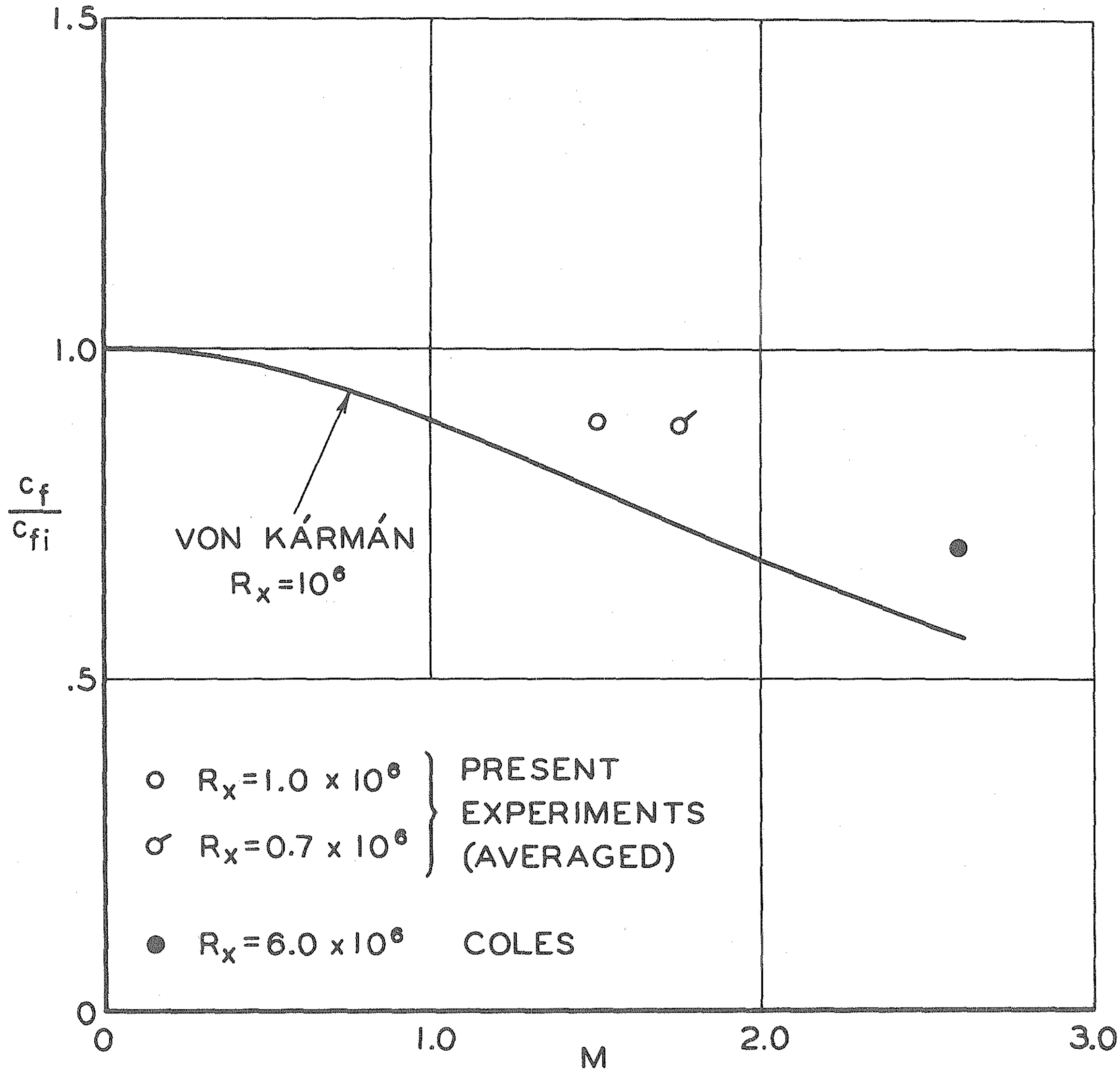

Figure 14.- Supersonic results. Incompressible results are from reference 9.

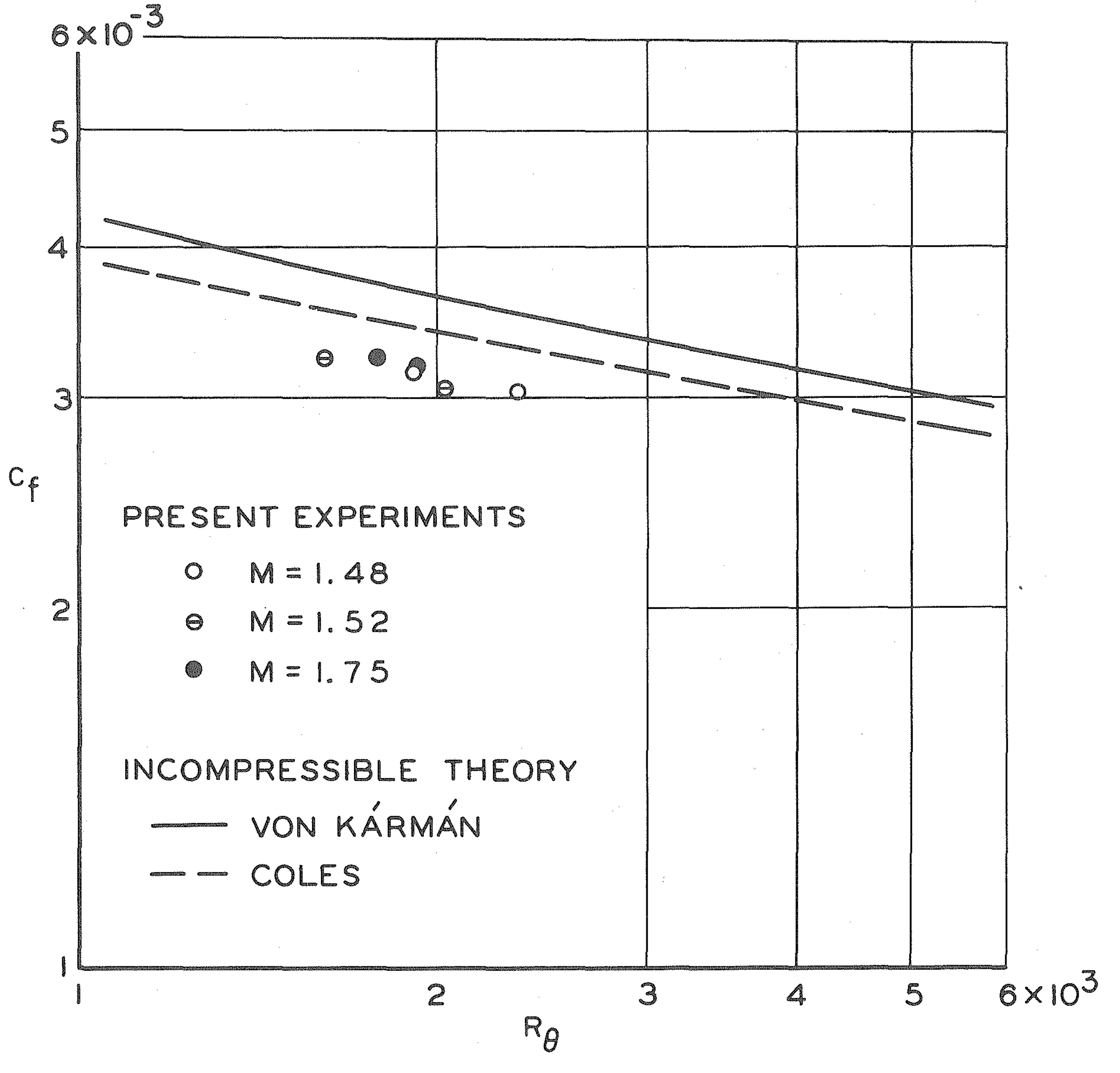

Figure 15.- Skin friction in terms of momentum thickness.

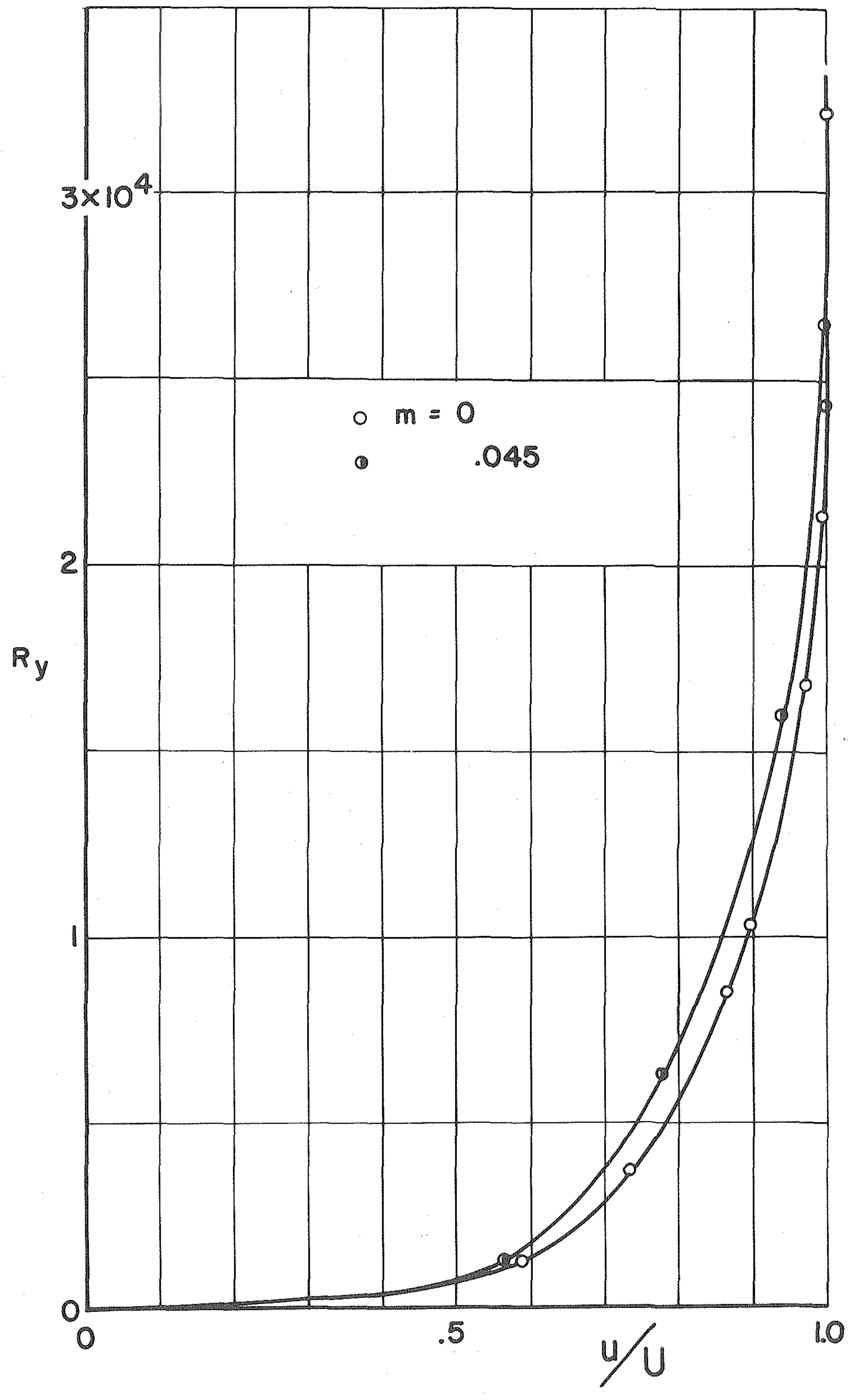

Figure 16.- Supersonic velocity profiles at $M = 1.48$.

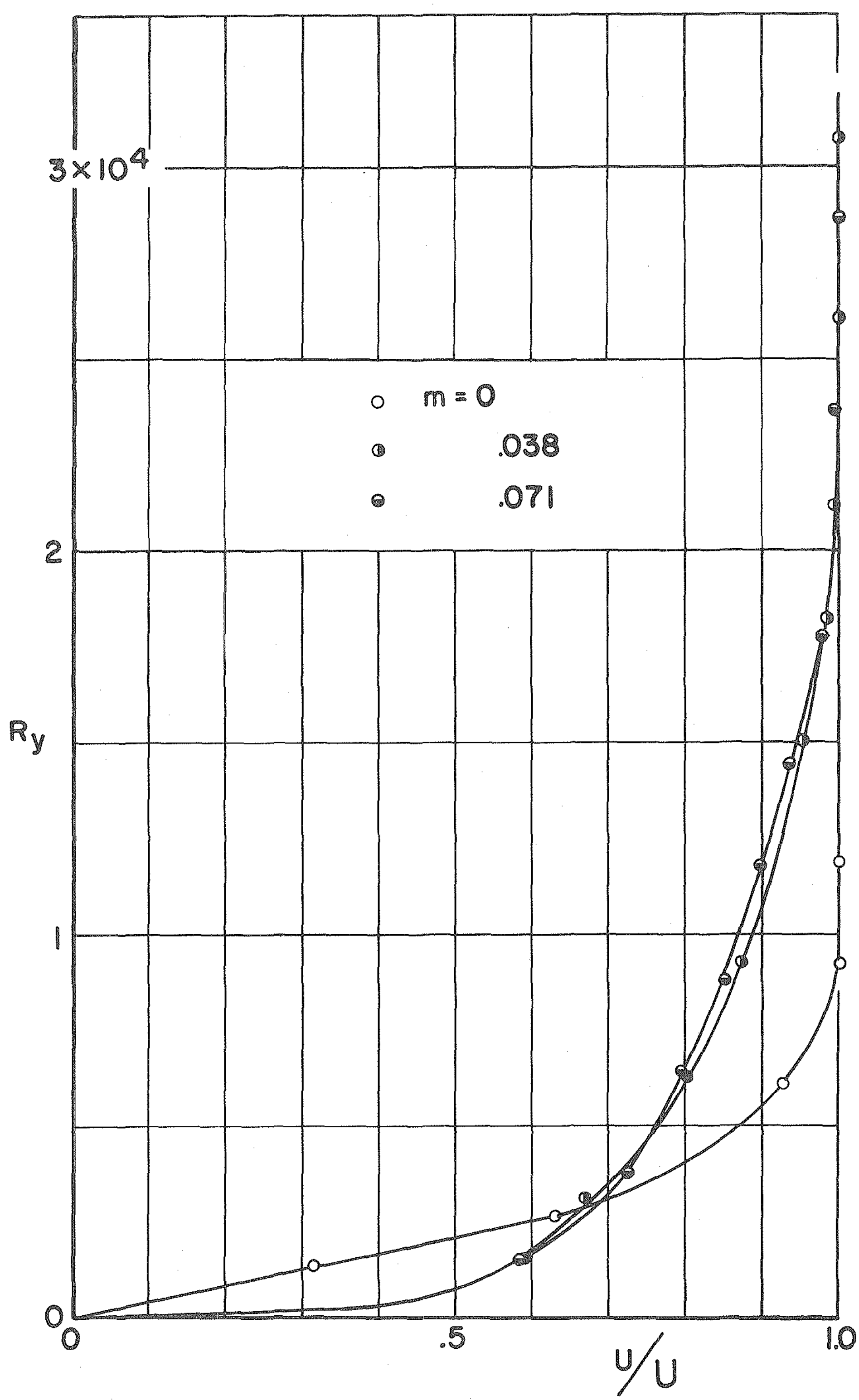

Figure 17.- Supersonic velocity profiles at M = 1.75.

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MEASUREMENTS OF TURBULENT SKIN FRICTION ON A FLAT PLATE AT TRANSONIC SPEEDS. Raimo J(aakko) Hakkinen, California Institute of Technology. September 1955. 41p. diagrs., photo., tabs. (NACA TN 3486)

The design and construction of a floating-element skin-friction balance are described. This instrument was applied to measurements of local skin friction in the turbulent boundary layer of a smooth flat plate at high-subsonic Mach numbers and supersonic Mach numbers up to 1.75. The principal difficulties which exist in comparing skin-friction coefficients at various Mach numbers are discussed.

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